

Overview of the Altair Lunar Lander Thermal Control System Design and the Impacts of Global Access

Ryan A. Stephan¹

NASA Johnson Space Center, Houston, Texas, 77058

NASA's Constellation Program (CxP) was developed to successfully return humans to the Lunar surface prior to 2020. The CxP included several different project offices including Altair, which was planned to be the next generation Lunar Lander. The Altair missions were architected to be quite different than the Lunar missions accomplished during the Apollo era. These differences resulted in a significantly dissimilar Thermal Control System (TCS) design. The current paper will summarize the Altair mission architecture and the various operational phases associated with the planned mission. In addition, the derived thermal requirements and the TCS designed to meet these unique and challenging thermal requirements will be presented. During the past year, the design team has focused on developing a vehicle architecture capable of accessing the entire Lunar surface. Due to the widely varying Lunar thermal environment, this global access requirement resulted in major changes to the thermal control system architecture. These changes, and the rationale behind the changes, will be detailed throughout the current paper.

Nomenclature

α	=	radiator solar absorptivity (unitless)
ε	=	radiator infrared emissivity (unitless)
E_{SHReD}	=	amount of energy required to be removed via sublimation (J)
h_{fg}	=	heat of vaporization for consumable feedwater (J/kg)
m	=	mass of consumed feedwater (kg)
q''_{solar}	=	incident solar flux (W/m^2)
q''_{IR}	=	incident infrared flux (W/m^2)
$Q_{SHReD}(t)$	=	time-varying heat-rejection requirement for the SHReD, shown as "green" curve in Figure 6 (W)
σ	=	the Stefan-Boltzmann constant ($W/m^2 \cdot K^4$)
T_s	=	radiative sink temperature (K)

I. Introduction

In response to President Bush's 2004 Vision for Space Exploration, NASA was planning a human return to the Moon prior to 2020. In preparation for this mission, NASA had established the Constellation Program (CxP). Within the CxP there were several project offices. One of these projects included the development of NASA's new Lunar lander vehicle, Altair. This paper will provide the reader with a very brief overview of the Altair vehicle and mission architecture. As implied by the title, special consideration will be given to the vehicle's Thermal Control System (TCS). The TCS requirements, functionality, and planned hardware will also be discussed.

II. Altair Overview

The planned Lunar mission architecture required the use of two separate launch vehicles, Ares I and Ares V. In addition to these critical assets, a successful Lunar mission was also designed to use an Earth Departure Stage (EDS), an Orion crewed vehicle, and the aforementioned Lunar lander. The larger of the two launch vehicles, Ares

¹ Thermal Systems Engineer, Crew and Thermal Systems Division, Mail Code EC-2.

V, was designed to first lift both the EDS and Altair into Low Earth Orbit (LEO). Shortly thereafter, the second launch vehicle, Ares I, would launch Orion into LEO with four crew members inside the vehicle. Orion and the Altair/EDS stack would rendezvous in LEO, and the EDS would propel the now-integrated stack on a trajectory to the moon. The EDS was planned to be jettisoned immediately after the trans-lunar injection burn.

The mission phase following the trans-lunar injection burn was referred to as the trans-lunar coast (TLC). The TLC duration was planned to be approximately three days. After Orion and Altair arrived at the moon, Altair would have been used to insert the vehicles into a 100 km, circular, Low Lunar Orbit (LLO). The mission design required the vehicles to spend approximately 24 hrs in LLO before Altair would separate from Orion and begin its descent to the Lunar surface. The entire descent phase was planned to take approximately four hours to complete.

Once Altair landed on the moon, the Lunar surface phase was designed to last for approximately seven days. Throughout this time, the crew would have performed up to seven extravehicular activities (EVAs). Each EVA would have included two crew members for a total of eight hours. Approximately 112 work-hours would have been spent outside of the pressurized cabin. Once the seven days had elapsed, the Ascent Module (AM) would separate from both the Descent Module (DM) and the airlock (AL). The AM would then carry the four crew members back to the awaiting Orion capsule where the vehicles would again rendezvous. This time the rendezvous would occur in LLO. Once aboard Orion, the Altair AM would have been jettisoned back to the Lunar surface. Orion's service module would then push the crew back to Earth where the crew module and the four crew members would safely land in the ocean.¹

As alluded to in the preceding paragraphs, Altair was designed to consist of three functional components (see Figure 1). The largest and most massive functional component is the DM. The DM is an unpressurized module and consists of composite struts and includes the Main Propulsion System (MPS). The MPS, which is composed of eight massive propulsion tanks, is required to perform the Lunar Orbit Insertion (LOI) burn, Lunar descent, and landing on the Lunar surface. Of the eight tanks, four contain liquid hydrogen and the remaining four tanks contain liquid oxygen. The propellant within these tanks is required to fuel the engine throughout the mission. The DM is powered using fuel cells.

The two smaller structures shown in Figure 1 are referred to as the AM and the AL. As indicated by the name, the AM is used to return crew members to the waiting Orion at the end of their Lunar surface stay. The overwhelming majority of the avionics and a small ascent engine are included in the AM design. The AM also serves as the habitable volume for the four crew members during descent, ascent, and the seven-day Lunar surface stay. The AM is nominally pressurized to 10.2 psia while docked with Orion and 8.3 psia when separated from Orion. During ascent, the AM is powered by on-board batteries. The AL, which is attached to the AM through a docking tunnel, is maintained at the same pressure as the AM throughout the entire mission except when it is depressurized for EVA ingress and egress. The AL is used to ingress and egress the vehicle during EVAs and is designed to accommodate two crew members per ingress/egress. The AL is depressurized and repressurized several times during Lunar surface operations. While the AM performs Lunar ascent, both the DM and the AL remain on the Lunar surface. After the AM rendezvous with Orion, the AM is jettisoned and returned to the Lunar surface.

As mentioned above, Altair is designed to take a crew of four to the Lunar surface and support operations for up to seven days. These mission requirements far exceed the Apollo capability of two crew members for a total of three days on the Lunar surface. In addition to these differences, another major difference is that Altair is required to perform the LOI burn. On Apollo, the command/service module was responsible for performing this burn. However, because Orion is being optimized for earlier LEO missions, the LOI burn will be performed by Altair, which results in a much heavier landing vehicle as compared to the Apollo design.

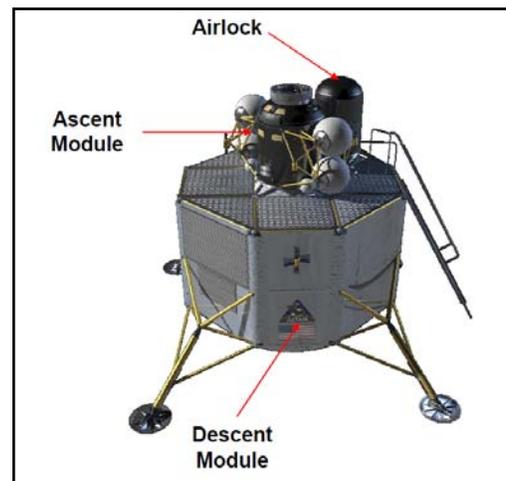


Figure 1. Altair lunar lander showing the three functional components: the AL, AM, and DM.

III. Thermal Control System Overview

From the simplest satellites to the most complex human-rated space vehicle, some degree of thermal control is required for all spacecraft. In its most basic form, thermal control is the maintenance of all vehicle surfaces and

components within an appropriate temperature range. For satellites, this requires that the TCS must maintain all of the equipment within its operating temperature range. Similar to the system for satellites, the TCS for human-rated space vehicles must also maintain all of the equipment within its operating temperature range. In addition to component-level temperature maintenance, the crewed spacecraft's TCS must also safely maintain the internal cabin temperature within appropriate temperature requirements to ensure crew survivability and comfort.

An effective TCS must provide three basic functions to the vehicle design: heat acquisition, heat transport, and heat rejection. Heat acquisition is the process of transferring thermal energy from a heat source to the TCS. The second critical function of a TCS is heat transport, which involves the movement of thermal energy from one region to another. The third and final function is heat rejection, which is the process of rejecting the excess thermal energy acquired by the TCS to the external environment.

IV. Altair Thermal Control System Heat-rejection Requirements

The first step in designing Altair's TCS is to understand the derived thermal system heat-rejection requirements. The system requirements are developed using knowledge of the vehicle's power requirements, the thermal environment, and the efficiency of the power generation system. For Altair, vehicle power is generated using either fuel cells or batteries depending on the mission phase. The vehicle's heat-rejection requirement varies throughout the Lunar mission as shown in Figure 2.

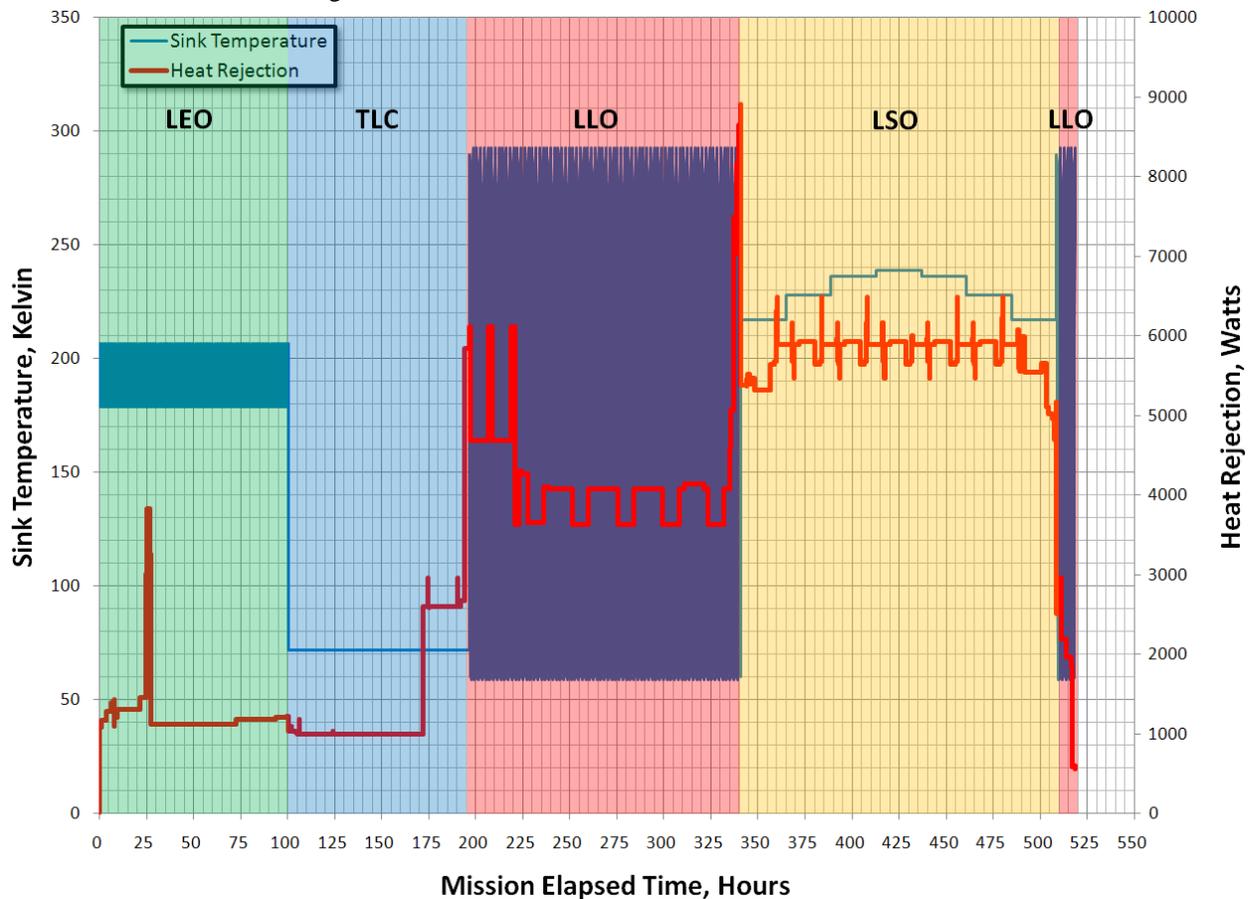


Figure 2. Altair heat-rejection requirement: LEO through completion of AM disposal.

In the preceding figure, the heat-rejection requirement is located along the right ordinate and the Mission Elapsed Time (MET) is shown along the abscissa. The ordinate shown on the left side of the plot represents the radiator sink temperature assuming optical properties representative of silverized Teflon[®] tape ($\alpha = 0.10$, $\epsilon = 0.85$). The instantaneous sink temperature is calculated using the following equation:

$$\text{---} \tag{1}$$

where: T_s = radiative sink temperature (K)
 α = radiator solar absorptivity (unitless)
 ε = radiator infrared (IR) emissivity (unitless)
 q''_{solar} = incident solar flux (W/m^2)
 q''_{IR} = incident IR flux (W/m^2)
 σ = the Stefan-Boltzmann constant (W/m^2-K^4)

The plot is color-coded by the vehicle's location in the mission profile. Starting from an MET of zero, the green region represents the period of time the vehicle is located in LEO. It should be noted that the short launch period (less than 15 minutes) is also included in this section of the plot. During LEO, the heat-rejection requirement is approximately 1,200 Watts for the majority of the mission profile. However, there is a small period of time during which the heat-rejection requirement increases to just under 4,000 W. This spike in the vehicle's heat-rejection requirement corresponds to the time when the vehicle is checking out several systems including the vehicle's fuel cells. The corresponding sink temperature shown in Figure 2 varies cyclically between 180 K and 207 K.

The following blue region is the portion of the mission in which the vehicle is performing the Trans-Lunar Coast (TLC). During this mission phase, the heat-rejection requirement is approximately 1,000 Watts before increasing to 2,600 Watts as the fuel cells are powered on and the vehicle prepares for the Lunar Orbit Insertion (LOI) burn. As shown in Figure 2, TLC is an extremely cold thermal environment. The radiative sink temperature is less than 75 K for the duration of this mission phase.

Unlike the previously described mission phases, the sink temperature in LLO varies significantly throughout the mission phase. As shown in Figure 2, the radiative sink temperature for a beta zero LLO is approximately 60 K when the spacecraft is shadowed by the moon. However, when the vehicle is located directly between the sun and the Lunar surface, the sink temperature is approximately 290 K. This wide variation in sink temperature results in a significant design challenge that will be discussed later in the current document. As is the case with the sink temperature, the heat-rejection requirement also varies substantially throughout this mission phase from a minimum value of 3,600 W to a maximum requirement of 8,300 W just prior to Lunar descent. The time-weighted average (used for subsequent thermal analysis) heat-rejection requirement during LLO is approximately 4,185 W.

The next mission phase, color-coded yellow in Figure 2, is Lunar Surface Operations (LSO). Throughout much of this mission phase, the heat-rejection requirement varies between 5,650 Watts and 6,500 Watts. The radiative sink temperature also varies over the course of the seven-day Lunar surface stay because the moon is rotating about its axis. The radiative sink temperature for a horizontal surface located at the equator varies between 217 K and 239 K. This sink temperature variation will be described in more detail later in this document.

During the short ascent phase, the heat-rejection requirement is between 2,500 W and nearly 3,000 Watts. The remaining mission phases (i.e., rendezvous with Orion, docked operations with Orion, and preparation for disposal) occur in LLO. During these mission phases, the heat-rejection requirement varies between approximately 600 and 2,965 W. The TCS must continue to function throughout the ascent module disposal. During this period of time, the ascent module heat-rejection requirement is only 570 Watts.

V. Altair Thermal Control System Architecture

Thermal control systems are typically designed for the highest heat-rejection requirement in the warmest *continuous* thermal environment. Figure 2 clearly shows that this scenario occurs during Lunar surface operations. As mentioned above, the heat-rejection requirement during LSO varies between 5,650 and 6,500 Watts and the corresponding sink temperature varies throughout the seven-day surface stay. It should be mentioned that the sink temperature shown in the preceding figure is for an equatorial mission straddling the subsolar point (i.e. Altair lands 84 hours prior to Lunar noon). This seven-day environment provided the worst-case warm thermal environment. A schematic of the Altair TCS design developed to meet the aforementioned system requirements is shown in Figure 3.

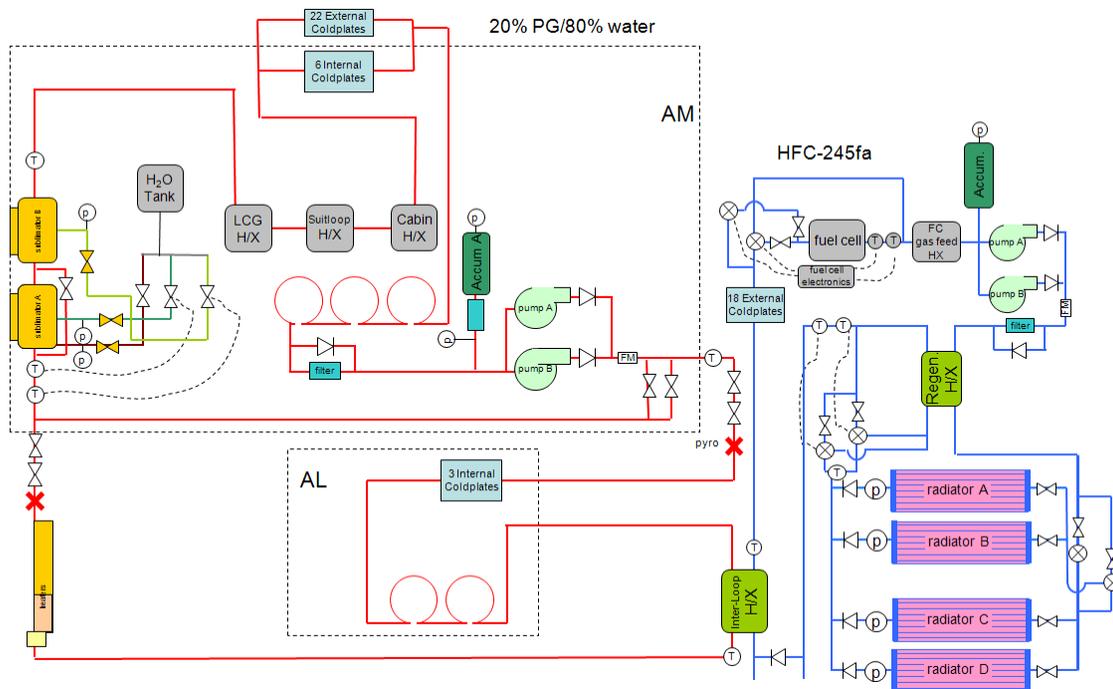


Figure 3. Altair thermal control system schematic.

Altair’s TCS accomplishes the heat acquisition function using various coldplates and heat exchangers, as shown in Figure 3. There are two different types of heat exchangers included in the system design. The first of these heat exchanger types is an air/liquid heat exchanger. As the name implies, this type of heat exchanger is used to transfer thermal energy from an air loop to a liquid cooling loop. The TCS design includes two air/liquid heat exchangers. The primary air/liquid heat exchanger, which is labeled “Cabin H/X” in Figure 3, transfers energy from the air in the AM to the TCS. The majority of this thermal load is from the crew metabolic load, but the thermal load also includes airborne energy from the various avionics located in both the AL and AM. The second air/liquid heat exchanger is used to acquire energy from a suited crew member. This heat exchanger is only used during mission phases in which crew members are in their spacesuits.

The second type of heat exchanger, a liquid/liquid heat exchanger, acquires energy from one liquid loop and transfers it to a second liquid loop. There are three liquid/liquid heat exchangers included in the system design. The first is the Liquid Cooling Garment (LCG) heat exchanger, the purpose of which is to acquire energy from the crew’s LCG and transfer it to the TCS. As before, this heat exchanger is only used when crew members are in their spacesuits. The second liquid/liquid heat exchanger, the inter-loop heat exchanger, is used to transfer energy from the internal to the external pumped fluid loop². The third, and final, liquid/liquid heat exchanger, the regenerative heat exchanger, is used to maintain the system setpoint throughout the various mission phases during which the radiators are being used.

There are a total of 49 coldplates included in the Altair TCS. The overwhelming majority (28 of the 49) are located on the vehicle’s AM. The coldplates are required to acquire excess thermal energy from various avionics components while maintaining these devices within their acceptable temperature limits. In addition to these coldplated items, there were four avionics boxes (the four AM RCS Drivers) that were conductively coupled to dedicated radiators. Because these items were not centrally located with the vehicle’s pumped fluid loop, it was more mass efficient to provide them with passive radiators than to run fluid lines to their location. These radiators are approximately 0.25 m x 0.31 m.

The second critical function of a TCS is heat transport. For the Altair system design, the heat transport function is accomplished using a two-loop system architecture. A two-loop architecture was required due to concerns about the fluid inside the radiator tubes freezing during TLC. As shown in Figure 2, the system heat-rejection requirement is very low during this mission phase (less than 1,000 Watts). This fact, combined with TLC being an extremely cold thermal environment, leads to concerns regarding fluid freeze. Unfortunately, fluids that have low enough freeze

temperatures (< 170 K) are typically toxic to the crewmembers, which is a concern in the event of an inadvertent fluid leak in the pressurized module. Therefore, safety concerns led to the selection of a nontoxic fluid for any loop located inside the AM or AL.

The resulting internal pumped fluid loop (depicted as the “red” lines in Figure 3) uses a mixture of propylene glycol and water as the working fluid. This particular fluid formulation was chosen to ensure commonality with the Orion system design. There was a strong desire within the CxPO for commonality across the program elements (Orion, Altair, and Lunar Surface Systems). The mixture of propylene glycol and water was also chosen because it has desirable thermal properties. In addition, this formulation provides a small level of freeze protection and is anticipated to be compatible with aluminum. This loop gathers thermal energy from the aforementioned heat exchangers and coldplates located on both the AM and the AL. Another unique aspect of the internal loop design is that it is used to provide condensation control for the vehicle. This function is accomplished by incorporating “omega loops” (the five omega-shaped figures in Figure 3). The omega loops are simply fluid loops that are thermally coupled to areas of the spacecraft that are particularly susceptible to condensation (hatches, windows, etc.).

The internal loop is driven by a centrifugal pump, and the system flow rate is approximately 300 lb/hr. Due to the criticality of the system (failure of the system would lead to loss of crew), a parallel redundant pump is included in the design. This secondary pump serves as a “cold backup” because it would only be activated if the primary pump failed at any time during the mission. Energy acquired by the internal loop is transferred to the external loop through the previously described inter-loop heat exchanger.

The external loop is also driven by a centrifugal pump, which is backed up by an identical inactive pump. Unlike the internal loop, failure of the external loop pumps would only result in loss of mission. However, due to the relatively low mass of a second pump, the vehicle engineering team decided to include the “cold backup”. The baseline working fluid for the external loop is HFC-245fa. This fluid was chosen due to its low freezing temperature and desirable mechanical properties. HFC-245fa is a hydrofluorocarbon that was developed as a replacement to the ozone-depleting chlorofluorocarbon. The flow rate for the HFC-245fa is a constant 1,270 lb/hr throughout the mission profile. Once the external loop acquires energy from the internal loop, it continues to service several coldplates and also provides cooling for the fuel cells while the fuel cells are operating. After acquiring energy from the fuel cells, the loop is also used to condition the fuel cell reactants prior to entering the power-generating device. The entirety of the acquired energy is then transported to the radiator panels where it is rejected to space.

The Altair TCS relies on two separate hardware components to achieve the heat-rejection function of the system. For the majority of the mission, heat rejection is accomplished using four deployable radiator panels located on the DM. These four radiator panels are located 90 degrees apart from one another and are insulated on the backside as shown in Figure 4. The backside is the side pointing toward the cryogenic tanks while stowed and toward the Lunar surface while deployed. The radiator panels were sized for Lunar surface operations, and the total surface area is approximately 28 m². To decrease manufacturing and flight costs, each of the panels are identical and measure 1.9 m wide by 3.7 m long. The radiator system mass includes the radiator panels and the mechanisms required for radiator deployment. The radiators remain stowed until Altair completes Lunar descent. After touchdown, the radiators are deployed into the horizontal position.

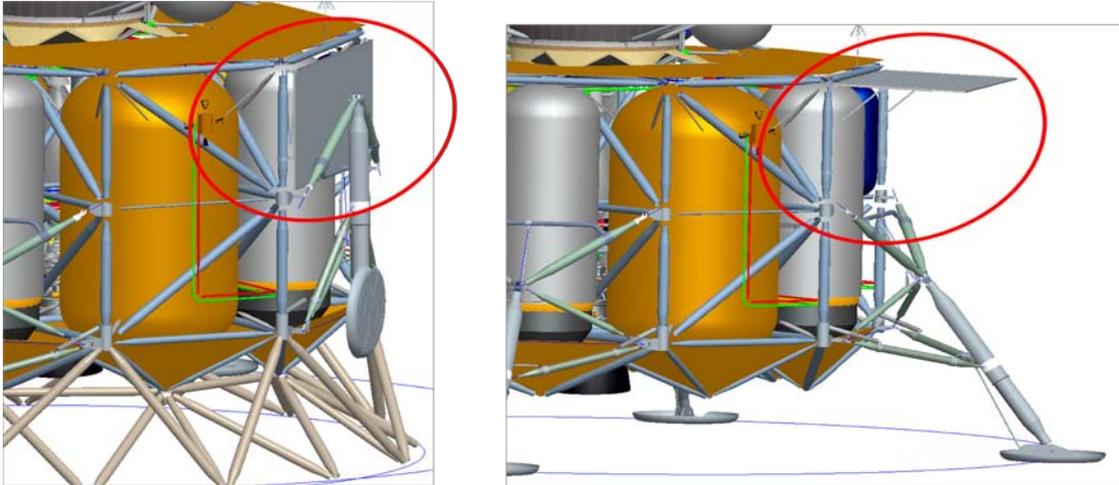


Figure 4. Vehicle descent module showing radiator and landing gear in stowed position (left) for launch and in the deployed position (right) after the vehicle lands on the Lunar surface.

The radiator panels are sufficiently sized to reject the full vehicle heat load throughout all of the mission phases until LLO operations. There are several instances during this mission phase in which the radiator's thermal environment is too warm for the radiators to reject the full vehicle heat load while maintaining the system setpoint. The warm thermal environment is due to the high IR backload incident on the vehicle's radiator while situated above the Lunar surface's sub-solar point. The sub-solar point is the point on the Lunar surface directly aligned with the sun. Figure 5 shows the spatial variation of temperature on the Lunar surface.

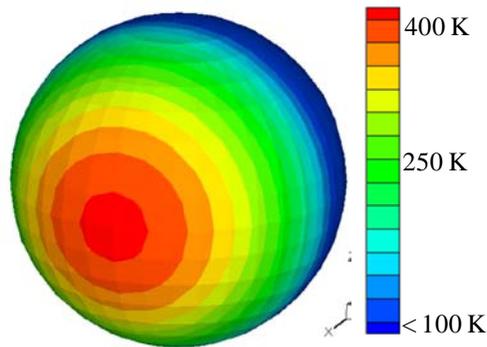


Figure 5. Spatial variation of the lunar surface temperature.

As shown in the preceding figure, the Lunar surface temperature at the sub-solar point reaches temperatures approaching 400 K. By contrast, the surface temperature on the cold side of the moon is less than 100 K. These wide variations in surface temperature result in a variable incident IR heat flux on an orbiting vehicle.

Using knowledge of the varying incident IR and solar loads, an instantaneous sink temperature can be calculated for a beta angle of zero degrees and an orbital altitude of 100 kilometers. The instantaneous sink temperature was calculated using Equation 1.

$$\text{---} \tag{2}$$

Where:

- T_{sink} Instantaneous radiative sink temperature (Kelvin)
- α Radiator surface solar absorptivity (Unitless)
- ϵ Radiator surface infrared emissivity/absorptivity (Unitless)
- q''_{solar} Incident solar heat flux (W/m^2)
- q''_{IR} Incident infrared heat flux (W/m^2)

σ Stefan-Boltzman constant ($\text{W/m}^2\text{-K}^4$)

The corresponding instantaneous radiator capability for these orbital parameters (thermal mass of the system was not considered for this analysis) is shown in Figure 6.

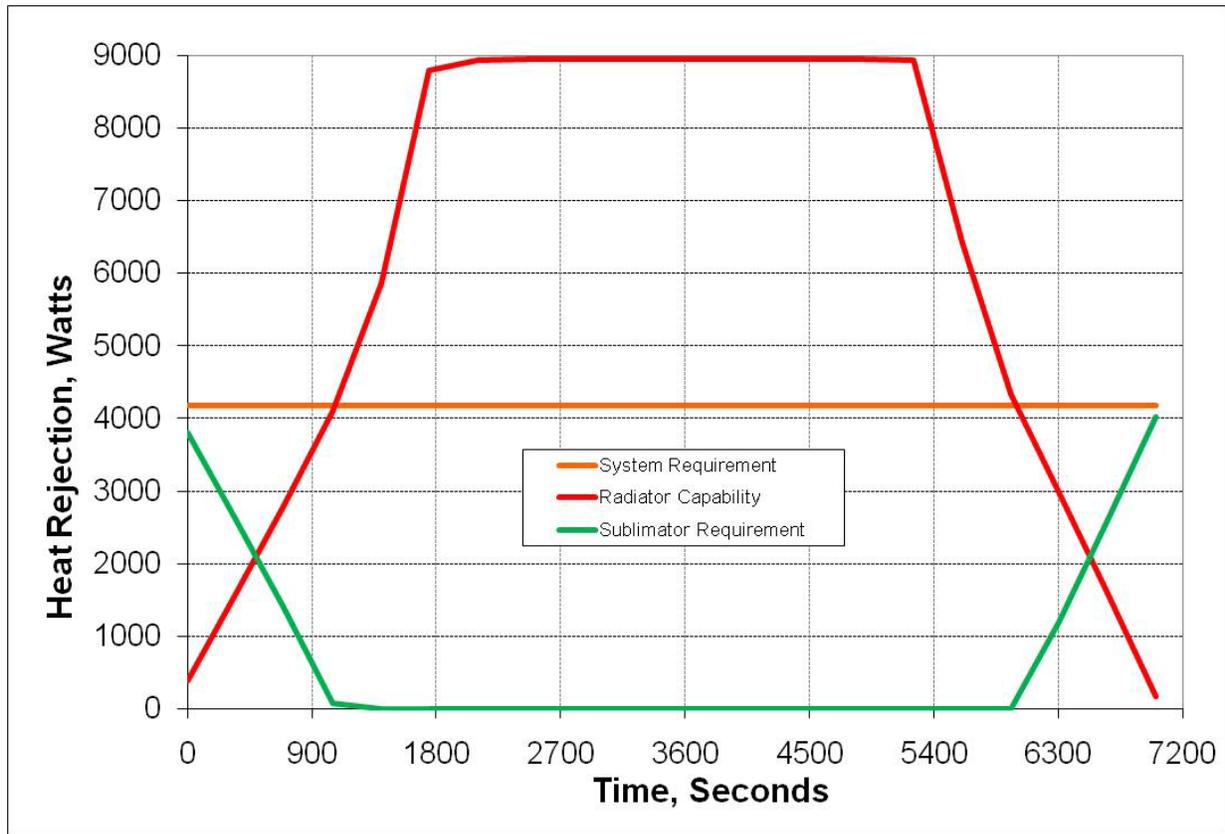


Figure 6. Radiator capability and the associated sublimator requirement for a circular 100-km lunar orbit.

In Figure 6, the vertical axis represents the heat rejection while the horizontal axis is the duration of a single Low Lunar orbit. Time zero corresponds to the time at which the vehicle is situated immediately above the sub-solar point. At this time, the radiator capability (shown in “red”) is quite low, given the relatively warm sink temperature associated with this orbital position. However, as the vehicle orbits the moon, the sink temperature decreases and the associated radiator capability increases to almost 8,000 W. As discussed above, the time-weighted average heat-rejection requirement during LLO is approximately 4,185 W (shown in “orange”). In studying Figure 6, it is apparent that there are several instances during which the radiator is incapable of rejecting the full vehicle heat load. At these times, a Supplemental Heat-Rejection Device (SHReD) is required. The SHReD requirement can be obtained by simply subtracting the radiator capability from the vehicle heat-rejection requirement. The resulting SHReD requirement is shown as the “green” curve in Figure 6. During the periods requiring supplemental heat rejection, a sublimator is used to reject the excess energy not rejected solely by the radiator panels. A sublimator was chosen as the SHReD because this type of hardware was already included in the system design for heat rejection during Lunar ascent and descent. In addition, a mass trade-off was performed and the addition of Phase Change Material heat exchangers was not warranted. The Altair TCS includes two identical sublimators. The second sublimator serves as a cold backup to be used in the event that the primary sublimator fails. Two sublimators were chosen due to the criticality of the system. The sublimator is the only means of heat rejection during ascent and failure of this hardware would result in loss of crew.

A sublimator is a type of evaporator that requires a consumable feedwater which is supplied to the sublimator where it freezes when exposed to space vacuum. The now-frozen feedwater layer sublimates to space when energy is applied by the warm coolant loop. The mass of feedwater required to perform this supplemental heat-rejection function for a single orbit was calculated using the following equation:

Where:

m_{LLO}	Mass of consumed feedwater in Low Lunar Orbit (kg)
E_{SHReD}	Amount of energy required to be removed via sublimation (J)
t_f	Duration of a Lunar orbit (sec)
$Q_{SHReD}(t)$	Time-varying heat-rejection requirement for the SHReD, shown as “green” curve in Figure 6 (W)
h_{fg}	Heat of vaporization for consumable feedwater (J/kg)

Using the previous equation, the amount of feedwater consumed during a single orbit is approximately 1.65 kg. However, because the sublimator is being used in a cyclical fashion (in accordance with the need as depicted in Figure 6), one must account for inefficiencies associated with this type of operational mode. Previous testing has shown that the sublimator efficiency may be around 88% when used in this manner³. Taking the sublimator efficiency (or utilization) into account results in a feedwater consumption rate of 1.87 kg per orbit. During LLO operations, the total sublimator water consumption was calculated to be approximately 114.6 kg.

During Lunar descent, a sublimator is used as the only means of heat rejection. This assumption was made due to uncertainty surrounding the thermal environment during Lunar descent. It is plausible that the thermal environment during descent is too warm for the radiators to reject the full vehicle heat load. To be conservative, the system was designed to only use a sublimator during this mission phase. A total of 12.8 kg of water is consumed during Lunar descent.

As mentioned above, the entire TCS, including the radiators, was sized for Lunar surface operations. In studying Figure 5, it is clear that the driving hot case for thermal control system design is an equatorial mission. The Lunar surface-stay duration is approximately seven Earth-days (168 hours). The driving design environment occurs when the mission “straddles” Lunar noon (i.e. 84 hours on the Lunar surface before the sun is directly overhead and 84 hours after the sun is overhead). Due to the high infrared backload incident upon a vertical surface for midday equatorial missions, a vertical body-mounted radiator cannot be used to reject energy while the system setpoint temperature is maintained. Resultantly, it was necessary to design the system to have horizontal radiators. The thermal control system design includes deployable radiators because there is not adequate surface area available on the vehicle to accommodate an adequately sized horizontal, body-mounted radiator.

At the request of vehicle integration office personnel, the radiators were sized to take advantage of the 125 kilograms of excess water available. This excess water is available because water is a byproduct of the fuel cells.

As shown in Figure 2, the Lunar surface heat rejection requirement varies between 5,650 Watts and 6,500 Watts. However, it is not necessary to size the thermal control system for the relatively quick transients lasting less than ½ hour shown in this figure. Resultantly, the TCS was sized for 5,988 Watts. As the moon rotates about its axis, the sink temperature for a horizontal surface varies from 217 Kelvin to 239 Kelvin at Lunar noon.

It was desirable to minimize the radiator area (hence the radiator mass) while maximizing the use of available sublimator consumable feedwater. To accomplish this design goal, a simple analytical process was followed. Using a methodology similar to that described above, the instantaneous sink temperature was calculated for the seven-day Lunar surface stay. With knowledge of this parameter, an instantaneous radiator capability was calculated for several different radiator surface areas. Using the heat rejection requirement of 5,988 Watts and the now-derived radiator capability, the transient radiator deficit was calculated. The radiator deficit (or sublimator requirement) was calculating using the following equation:

$$Q_{\text{deficit}} = Q_{\text{requirement}} - Q_{\text{radiator, capability}} \quad (4)$$

Where:

Q_{deficit}	Heat rejection requirement by the sublimator (Watts)
$Q_{\text{requirement}}$	Altair heat rejection requirement (Watts)
$Q_{\text{radiator, capability}}$	Instantaneous radiator heat rejection capability (Watts)

The radiator deficit, or heat rejection requirement for the sublimator, is shown for several different radiator area assumptions in Figure 7.

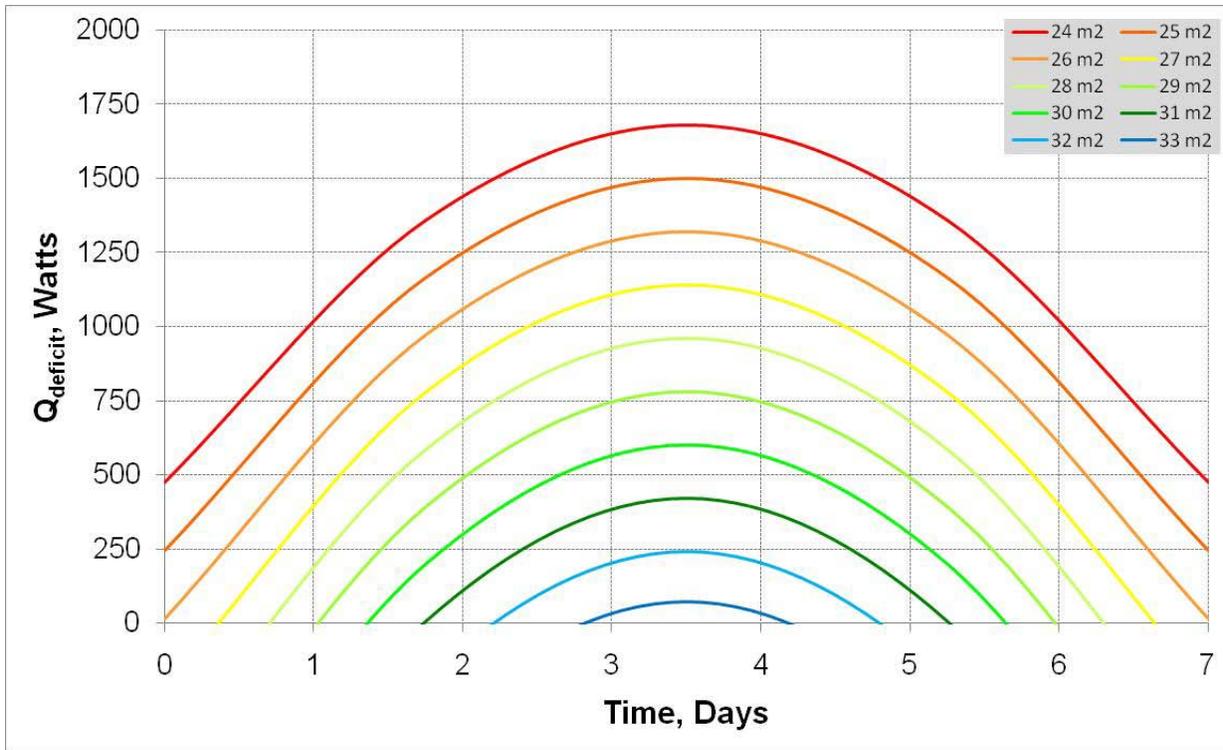


Figure 7. Time-varying sublimator heat rejection requirement during seven-day Lunar surface mission.

The data from the preceding figure can be used to calculate the mass of water required during the seven-day Lunar surface stay using the following equation:

$$\text{---} \quad (5)$$

Where:

$m_{\text{Lunar Surface}}$ Mass of water required by the sublimator during Lunar surface stay

The mass of water required for sublimation on the Lunar surface as a function of radiator surface area is shown in Figure 8.

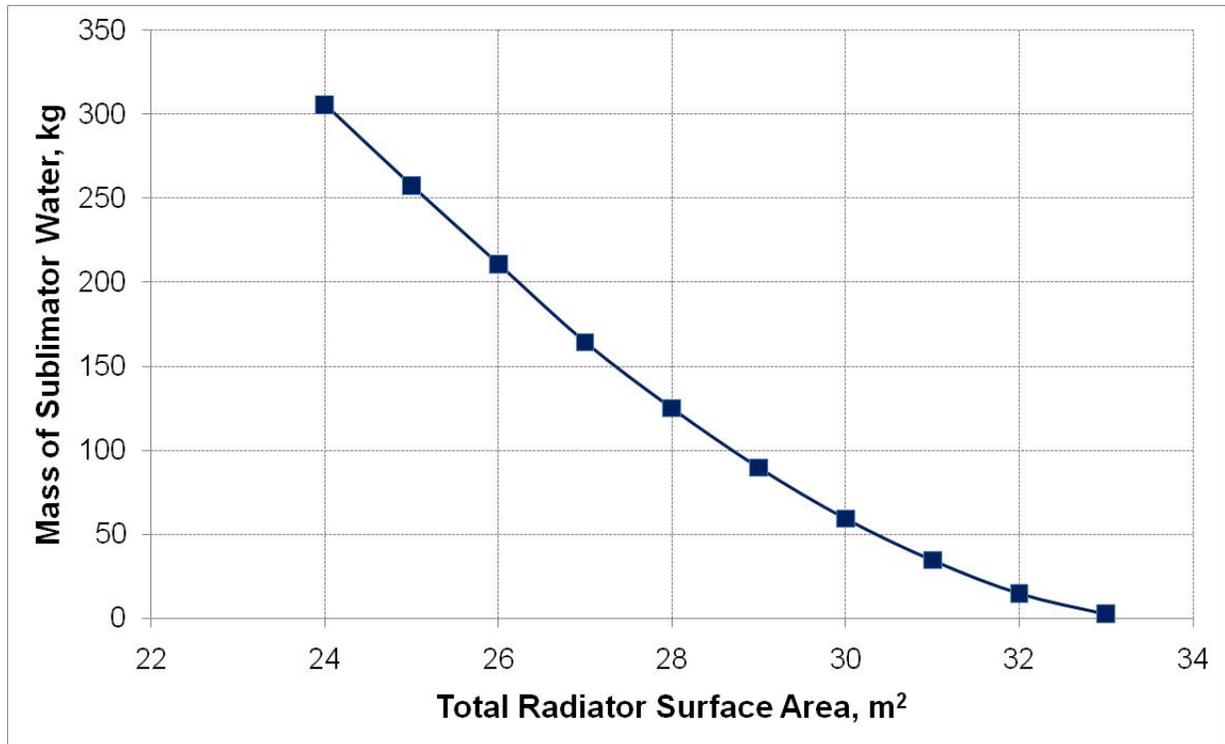


Figure 8. Mass of sublimated water versus total radiator surface area for Lunar surface operations.

As mentioned above, the thermal control system was designed to use the smallest possible radiators while consuming the 125 kg of excess water available while situated on the Lunar surface. In studying Figure 8, it is clear that a thermal control system with 28 m² of active radiator surface area consumes approximately 125 kg of water via sublimation. Therefore, this area was chosen for the radiator design as described above.

Because the AL and the DM are left on the Lunar surface during ascent, two pyrotechnic valves are included in the TCS design. These valves are shown as red x's in Figure 3. Immediately prior to ascent, several isolation valves are configured to isolate the internal TCS for ascent. These valves are used to bypass the AL and only direct the pumped fluid flow through the AM. After configuring these valves, the pyrotechnic valves are used to sever the internal pumped fluid loop lines. Because the radiator panels remain on the Lunar surface with the DM, the sublimator provides the heat-rejection function throughout Lunar ascent, Orion rendezvous, docked operations, and vehicle disposal. Figure 9 shows the required sublimator feedwater mass for each of the mission phases during which a sublimator is used as either supplemental or primary heat rejection.

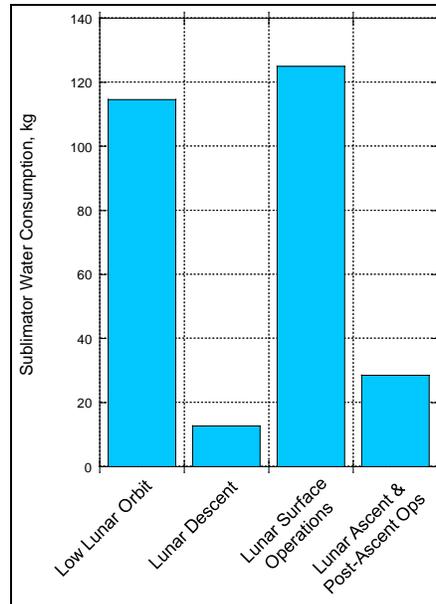


Figure 9. Mass of sublimator feedwater used throughout Altair’s mission profile

A total of approximately 281kg of water is required to meet the vehicle’s heat-rejection needs during the Altair mission. The overwhelming majority of that water is consumed during LLO and Lunar surface operations when the sublimator serves as a supplemental heat-rejection device. The other mission phases that require water to be sublimated are Lunar descent, Lunar ascent, and post-ascent LLO operations with Orion.

VI. Altair Thermal Control System Conclusion

The Altair thermal control system presented in the preceding sections was designed to effectively operate through a wide range of thermal environments including Low Earth Orbit (LEO), Trans-Lunar Coast (TLC), Low Lunar Orbit (LLO), and Lunar Surface Operations (LSO). The TCS was designed to meet the vehicle’s derived heat rejection requirements. These transient requirements were generated with knowledge of the vehicle power requirements, thermal environments, and the crew metabolic load requirements. One of the most challenging aspects for the thermal control system design is the requirement to operate in a warm environment with a high heat rejection requirement (occurring during LSO), but also maintain the system setpoint during the cold environment occurring during TLC. The system turndown ratio (defined as the ratio between high and low dissipation requirements) is approximately six to one, which is much more severe than Apollo. This design consideration led to a two-loop TCS architecture.

The thermal control system’s heat acquisition function is accomplished using several heat exchangers and coldplates. As mentioned above, a two-loop architecture using both a mixture of propylene glycol and water as well as HFC-245fa is employed to transport the acquired thermal energy to the heat rejection system. The heat rejection function is accomplished primarily using four deployable radiators. A unique design feature of the heat rejection system is that the radiator size was optimized for the approximately 125 kg of water available during LSO. In addition to using the sublimator as a supplemental heat rejection device during LSO, the sublimator also performs this function during LLO.

Acknowledgements

The author would like to thank Dr. Eugene Ungar for his insight and contributions to the original TCS design. The author would also like to thank Greg Schunk for providing timely and accurate thermal analyses. In addition, he would also like to thank Dr. Tom Leimkuehler for being there to provide guidance and to serve as a “sounding board” for various technical issues that have come up throughout the development of this design.

References

¹Polsgrove, T., Button, R., and Linne, D., "Altair Lunar Lander Consumables Management." *AIAA Space 2009 Conference & Exposition*. September 2009.

²Ungar, E. K., "Spacecraft Radiator Freeze Protection Using a Regenerative Heat Exchanger with Bypass Setpoint Temperature Control," SAE International Conference on Environmental Systems. San Francisco, California. July 2008.

³Leimkuehler, T. O., Sheth, R., and Stephan, R. A., "Investigation of Transient Sublimator Performance." SAE International Conference on Environmental Systems. July 2009.

Acronyms

AL:	airlock
AM:	ascent module
CxP:	Constellation program
DM:	descent module
EDS:	Earth departure stage
EVA:	extravehicular activity
IR:	infrared
LCG:	liquid cooling garment
LEO:	low-Earth orbit
LLO:	low-Lunar orbit
LOI:	Lunar orbit insertion
LSO:	Lunar surface operations
MET:	mission elapsed time
MPS:	Main propulsion system
RCS:	Reaction control system
SHReD:	supplemental heat-rejection device
TCS:	Thermal control system
TLC:	trans-Lunar coast